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The Orbital Design of Alpha Centauri Exoplanet Satellite (ACESat)

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ABSTRACT

Exoplanet candidates discovered by Kepler are too distant for biomarkers to be detected with foreseeable technology. Alpha Centauri has high separation from other stars and is of close proximity to Earth, which makes the binary star system 'low hanging fruit' for scientists. Alpha Centauri Exoplanet Satellite (ACESat) is a mission proposed to Small Explorer Program (SMEX) that will use a coronagraph to search for an orbiting planet around one of the stars of Alpha Centauri. The trajectory design for this mission is presented here where three different trajectories are considered: Low Earth Orbit (LEO), Geosynchronous Orbit (GEO) and a Heliocentric Orbit. Uninterrupted stare time to Alpha Centauri is desirable for meeting science requirements, or an orbit that provides 90% stare time to the science target. The instrument thermal stability also has stringent requirements for proper function, influencing trajectory design.

INTRODUCTION

The existence of a habitable exoplanet is of particular interest to the scientific community. Kepler's exoplanet discoveries have highlighted the possibility of life existing beyond Earth. These targets, however, are too distant for habitability to be measured with foreseeable technology. An Earth-like planet (at least a non-transiting one) must be directly imaged for an atmosphere to be established and for scientists to understand elemental composition. A coronagraph is an observational instrument that enables direct imaging to measure the spectra of an exoplanet by blocking out the light of the host star. The star system Alpha Centauri is not only of close proximity to Earth, but is also much closer than any other Sun-like star. Recent Kepler data has estimated as much as 40–50% chance of a Sun-like star supporting a habitable exoplanet.

Missions such as Transiting Exoplanet Survey Satellite (TESS) and James Webb Space Telescope (JWST) also can observe exoplanet targets, but their observational methods of transit photometry and spectroscopy are statistically unlikely to measure the spectrum of Earth-like planets. A proposal to use a coronagraph to observe both stars of Alpha Centauri called ACESat (Alpha Centauri Exoplanet Satellite) was submitted in response to a Small Explorer Program (SMEX) call.

ACESat is a <300 kg secondary payload mission that proposes to look at both Alpha Centauri stars in search for an exoplanet (Belikov et al., 2015). For a mission such as this to be successful, every spacecraft subcomponent (communications, power, propulsion, ADSC, and thermal) must work in harmony. The driving parameter for mission success centers on the pointing stability of the instrument, with a less than 10 arc second maximum deflection requirement. Secondly, the instrument has a stringent thermal constraint that requires a stable thermal environment. In order for the trajectory to satisfy these requirements, three different trajectories, Low Earth Orbit (LEO), Geosynchronous Orbit (GEO) and Heliocentric Orbits, were analyzed to evaluate which would support a more advantageous science mission. The primary program used for all orbit simulations was Systems Tool Kit (STK); MATLAB was also utilized for heliocentric orbit design analysis.

While this orbit list is not exhaustive, the presented orbit analyses must comply with necessary SMEX budget and design limitations. Trajectories rejected as a result of this analysis are due to the team not being able to close the mission with sufficiently low risk during the allocated submittal timeframe.

This paper will be presented as follows: first a discussion of the different orbit options available through commercial launch ride share options, then a presentation of the three different orbits considered followed by a description of how the final orbit was selected. Lastly, the improvements or alternatives to the baseline orbit will be explored.

ORBIT TRADES

Orbit Providers

Since ACESat will be able to hitch a ride as a secondary on a commercially available rocket, orbit selection is limited to what is accessible. Orbit providers Spaceflight Services, Orbital Sciences and Space Systems Loral (SSL) were the only vendors inquired for orbit selection for this mission; Table 1 describes the available trajectory options (inclination is abbreviated *i*). As a secondary payload, the propulsion system can be smaller which is beneficial for a smaller spacecraft mass, leaving additional room for other hardware. For this mission, a propulsion system will introduce perturbations to the instrument pointing, which needs to be avoided. If ACESat can enter an orbit that does not require station keeping or correction maneuvers, the on-board propulsion system can be relatively small or completely eliminated.

Table 1: Orbit Providers and Available Orbits

Commercial Launches	Orbit Type	Altitude (km)	*i* (deg)
Spaceflight Services	LEO	500-600	97.8
	LEO	500-600	63.4
	LEO	600-830	97.8
	LEO	600	52
	LEO	500-600	97.8
	LEO	500-600	63.4
	LEO	500-600	44
Space Systems Loral	LEO	450	97.8
	GEO	35200	0
	GTO	35786 x 300	28.5
Orbital Sciences	GEO	35200	0
	GTO	35786 x 300	28.5

Although the listed launch providers all have access to Geosynchronous Earth Orbit (GEO), the spacecraft would have to be released into Sub-GEO (~500 km below GEO altitude of 35,756 km) as a free flyer. This is to ensure that no communication satellites are disturbed. The Geosynchronous Transfer Orbit (GTO) is a highly elliptical orbit with a perigee at LEO parking orbit and apogee reaching the GEO belt.

Orbit Options

Each available orbit has its own positive and negative qualities that need evaluation. An orbit trade study was performed to determine what trajectory option is most beneficial to the ACESat mission, see Table 2 below.

Table 2: Orbit Trade Study

	LEO	GEO	Heliocentric
Pointing	50%	90%	100%
Thermal	Not stable	Moderate	Stable
Accessible	Yes	Yes	Yes
Propulsion	No	Yes	Yes
Data Rate	Low	Low	High
Radiation	~3 mm	~6 mm	~4 mm

As a secondary payload, we need to illustrate all launch opportunities of the primary are satisfied. Depending on what time of day the primary decides to be launched corresponds to particular Right Ascension of Ascending Node (RAAN). Therefore, the following orbit analyses include determining how the launch window is satisfied.

LEO

There are several benefits to being in a Low Earth Orbit (LEO) for this proposal. It is most accessible, lowest cost and none of the LEOs listed in Table 1 require any station keeping to maintain. Spaceflight Services was the only launch provider offering different rides to different LEOs; therefore these are the only LEO options considered.

The initial orbits analyzed were circular LEOs between 500–700 km at i = 44, 52 61, 63.4 and 97.8 deg. ACESat would ride up and be released as a free flyer into one of these available orbits. At these relatively low altitudes, when the Earth occludes Alpha Centauri, the instrument is looking into atmosphere. The atmospheric particles can potentially disturb the coronagraph and create perturbations. A grazing angle constraint of 30 deg is placed on the instrument in STK to eliminate this occurrence, see Figure 1.

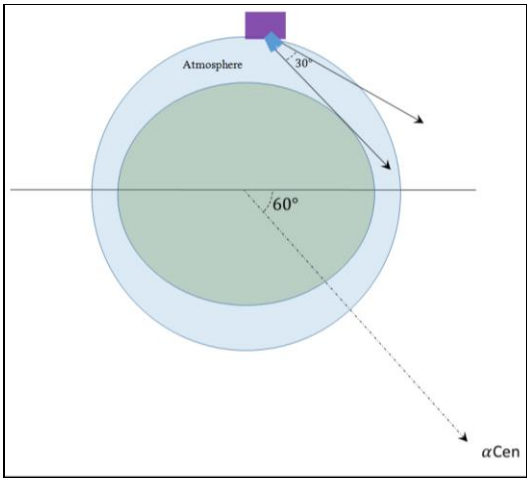

Figure 1: Grazing angle of 30 deg to eliminate atmosphere particles when not looking at Alpha Centauri (αCen).

These circular orbits are associated with periods ranging from 94-100 min with an average of 15 orbits per day. Due to the amount of times ACESat will orbit the Earth in one day, a minimum of 50% stare time per orbit to Alpha Centauri is accepted for minimal science. Figure 2 below graphs average stare time per orbit to the available LEO altitudes at four available inclinations. The arrows identify a minimum of 45 min per orbit of access time is attained at a minimum LEO altitude of 600 km. This means that ACESat cannot tolerate a LEO with <600 km altitude.

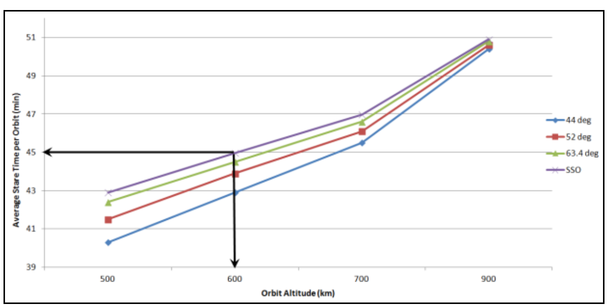

Figure 2: Average stare time per orbit vs available circular LEO.

The possible launch opportunities available per orbit are listed in Table 3, which include the grazing angle constraint. The last row of this table supplies the average stare time per orbit for every 45 deg RAAN. While there is a possibility of having 100% stare time if launched with a 135 deg RAAN, the chance of that happening is small and cannot be relied upon.

RAAN	600, *i*=44°	600, *i*=52°	600, *i*=63.4°	600, SSO	700, SSO	800, SSO
0°	44.8	45	45.1	44.8	46.9	48.8
45°	44.1	44.5	44.8	45.2	47.1	49
90°	41.4	42.6	43.7	45	47	48.9
135°	35.9	40.1	42.7	44.9	46.9	48.9
180°	42.4	43.2	44.1	45.1	47.1	49
225°	44.4	44.7	44.9	45.1	47.1	49
270°	44.9	45	45.1	44.7	46.8	48.8
315°	45	45.1	45.1	44.3	46.5	48.6
Mean	42.9	43.9	44.4	44.9	46.9	48.9

Table 3: Average stare time (min) to Alpha Centauri for one orbit over all possible RAAN values (increments of 45 deg) for available circular LEOs.

The duration when the ACESat cannot stare at Alpha Centauri is due to Sun eclipses. These produce an unstable thermal environment every orbit to the instrument which will constantly cause perturbations. Additionally, Earth's albedo will reflect sunlight into the coronagraph. These are consequences of the satellite having close proximity with Earth and are challenging to overcome with allocated SMEX budget and proposal submittal timeframe. These limitations can be further explored for future research to ensure low risk for this orbit option.

GEO

The Geosynchronous Earth Orbit (GEO) altitude of 35,786 km will offer continuous line of sight to Alpha Centauri every orbit throughout the entire mission. Due to the popularity of this orbit for communication satellites, most commercial orbit providers offer a ride to GEO. Here we can either be released into sub-GEO (as previously explained) as a free flyer or remain on-board a communication satellite as a hosted payload. SSL, a company that designs communication satellites, has offered to carry ACESat on one of their L1300 satellites launching in the same time frame.

As a hosted payload, there are several subsystem benefits: communications could be performed via 'mothership', propulsion or ADCS would not be needed and we would be allotted space on the mothership's power system. However, the position of ACESat would have to be next to the solar panel on the -Z side (see Figure 3). This Figure displays the labels of all faces of the SSL satellite modeled in STK. This is the only available location for ACESat as it needs to be far from the propulsion system, so ACESat's pointing is not perturbed, as well as from the antennas.

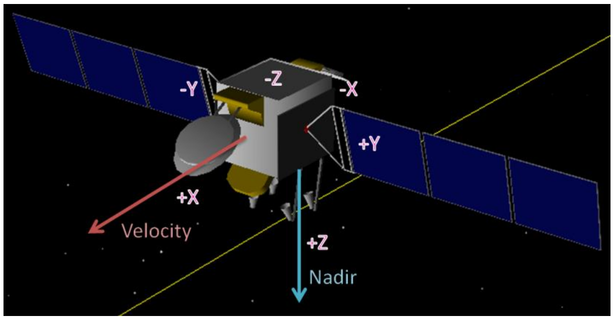

Figure 3: Axes for SSL GEO Communication satellite modeled in STK; ACESat would be located on –Z face in –Y direction.

However at this particular location, the solar panel of the SSL satellite obscures the boresight of the instrument during the autumn and winter months. During the time when the panels are not blocking the instrument view (spring and summer), sun light has the potential to leak into the lens of the coronagraph. This would create challenging perturbations to the instrument. A 5.15 m boom would facilitate this problem, however to design a stable boom that long and stable is over budget. This eliminates the GEO hosted payload opportunity for ACESat.

The free flyer option in GEO allows ACESat to orient itself in any manner that has 100% stare time to Alpha Centauri. The primary limitation here is the thermal instabilities generated from Earth's albedo. Secondly, this orbit requires a small propulsion system for decommissioning.

HELIOCENTRIC ORBIT DESIGN

Multiple trajectory iterations demonstrate that a heliocentric solar orbit would be most beneficial for the ACESat mission as this trajectory satisfies both the science and thermal requirements. As ACESat orbits the sun, it is able to stare uninterruptedly at Alpha Centauri, inclined 60 deg below the ecliptic plane, for the entire mission duration. At this location, there are no orbiting bodies to eclipse the spacecraft. This also introduces a stable thermal environment for spacecraft as it will not experience eclipses or endure Earth albedo effects. While there is no station keeping needed throughout the mission, an orbit insertion maneuver is required. STK simulations show that many solar orbit can be achieved with a single insertion maneuver at perigee of GTO of 800 m/s. Additionally, once the mission is complete, the spacecraft will already be in a disposal orbit and will not need extra propulsion for decommissioning.

Being a secondary payload is the main disadvantage, where all possible launch opportunities need to be satisfied. The time of day the primary launches will affect the escape trajectory. The second disadvantage is how the distance between the spacecraft and Earth increases throughout the analyzed three year time frame, which will constrain how the communications subsystem is designed. Due to the communication budget limitations, the maximum allowable drift after the three year mission is 0.5 AU.

Launch Opportunity

Since ACESat will be released into a GTO, we need to understand the launch availability as a secondary payload. Figure 4 below illustrates the different RAAN values associated with GTO for one day in Fall 2020. Depending on what RAAN value the primary chooses will affect the escape energy required for ACESat to escape Earth.

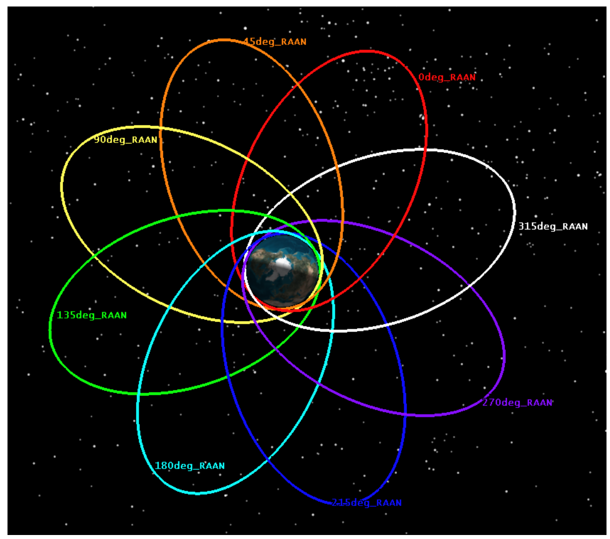

Figure 4: Range of GTO orbits for one day in Fall 2020 in 45 deg RAAN increments.

To see how many available escape trajectories can be obtained utilizing the drift rate and 800 m/s single orbit insertion maneuver constraint, a simulation in MATLAB was run. Figure 5 displays the resultant escape orbits, where the highlighted portion satisfies both requirements. The highlighted section in the graph represents a 12 hour period, which indicates that ACESat has a 50% launch window.

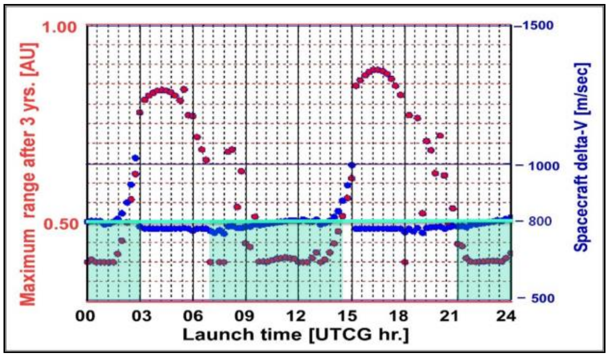

Figure 5: Launch window for escape trajectories for ACESat Earth-range and delta-V requirement.

In the trajectory design for the Space Infrared Telescope Facility (SIRTF), J. H. Kwok describes the dynamics for the heliocentric orbit injection design and found two classes of escape orbits that provide minimal Earth-spacecraft range: Earth Leading and Earth Trailing Orbits. Objects in an Earth Leading Orbits (ELO) will lead the Earth, while Earth Trailing Orbits (ETO) trails behind the Earth. In this injection design, Kwok describes the geometry for each ELO and ETO; an injection point at midnight results in an ELO, while noon (Sun side) injections result in ETO. This is demonstrated in Figure 5, where midnight–noon corresponds to an ELO and noon–midnight represents an ETO.

By graphing the Earth-ACESat range after three years over all possible RAAN values for one day, the different escape trajectories are illustrated in Figure 6 below. The two sets highlighted escape orbits are in RAAN sweet spots, where both constraints are satisfied. These RAAN sweet spots will enable the spacecraft to escape and not drift farther than 0.5 AU using <800 m/s delta-V. The first range, 100–180 deg, is associated with ETO and second, 280–360 deg, correspond to ELO. Here, 10 deg RAAN equates to a midnight launch and 190 deg RAAN represents a noon launch.

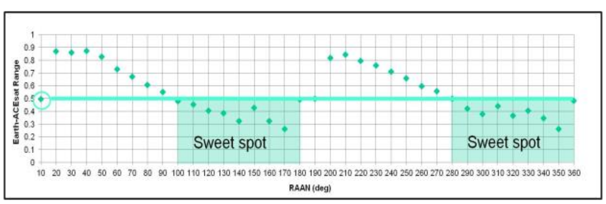

Figure 6: RAAN sweet spots are located to satisfy ACESat-Earth range requirement of 0.5 AU after three years.

If ACESat is injected into one of these orbits, all mission requirements will be satisfied. Worst case scenario where RAAN sweet spot does not match the primary is analyzed in the next subsection to ensure the integrity of the mission.

RAAN Optimization

The delta-V costs to escape Earth vary due to the third –body perturbations of the Moon and Sun and create a non-spherical boundary. As certain areas are much closer to Earth than others in this boundary, there are varying delta-V costs for a spacecraft to escape. A low energy transfer orbit referred to as Weak Stability Boundary (WSB) allows a spacecraft to change its orbit.

Worst case scenario for ACESat is starting with a RAAN value not in a sweet spot; the orbit insertion maneuver would not provide enough energy for the spacecraft to meet range requirements after three years. In this situation, ACESat can change its RAAN so an optimal escape trajectory is available by orbiting in a WSB. Due to Earth's rotation around the sun, the RAAN will slowly vary over the course of a year. Every three months the RAAN changes ~90 deg. If ACESat needed a 90 deg change in RAAN, it would take three months.

Figure 6 showed two RAAN sweet spots for ACESat's requirement for Earth range and delta-V can be depicted as four WSB quadrants in Figure 7 below. Quadrants II and IV contain the desirable escape trajectories for ACESat's requirements.

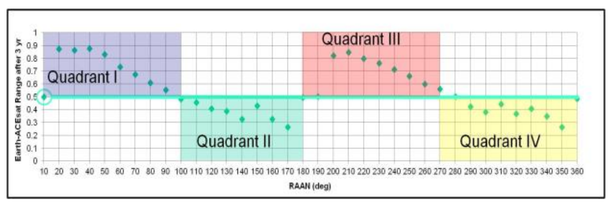

Figure 7: Four WSB quadrants are shown for the different escape orbits.

Again, this information can be further displayed as a circle modeled in STK. Figure 8 below shows the four RAAN quadrants and a way for the spacecraft to change the RAAN value of its escape orbit. Spacecraft do not ideally sit in GTO due to high radiation exposure, unless that is the purpose. Instead of staying in GTO during this time, ACESat would raise the apoapsis to 750,000 km by performing a burn at the perigee as shown in Figure 8. Since raising the apogee to that distance requires a lot of energy, the delta-V would use the majority of the propellant (730 m/s). Once the right escape trajectory is available, the remaining propellant can be used for escaping. Therefore no extra propellant is required for this method of achieving a desirable escape RAAN.

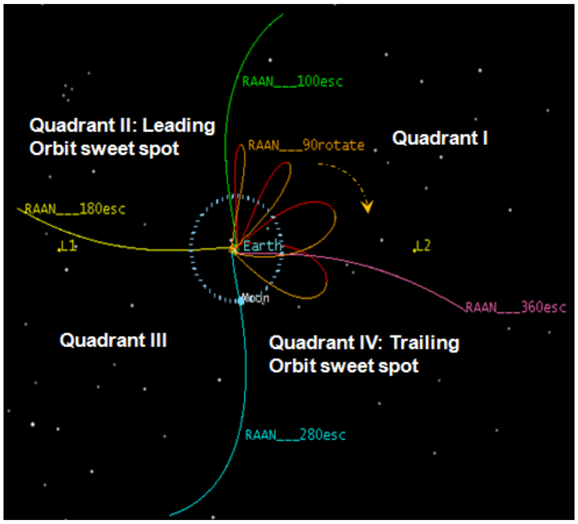

Figure 8: A spacecraft can allow of RAAN optimization to get into a different RAAN quadrant.

During this time, ACESat would naturally orbit the Earth for three months until a desired value of RAAN is obtained (Genova, 2014). ACESat can still perform science the months spent in Earth orbit waiting. There would be 27 days of continuous access to Alpha Centauri per orbit and experience a partially stable thermal environment once the spacecraft left perigee every month.

ORBIT OPTIMIZATION

There are a few optimization methods this orbit design could undergo, as this is constrained to the SMEX call it was proposed to. Instead of performing a single burn at perigee of GTO, ACESat could execute multiple burns over the course of the mission. This would raise and lower and apoapsis and periapsis of the heliocentric orbit thereby limiting ACESat's drift rate; ACESat would stay fairly close to Earth throughout the mission duration. However as mentioned, this entails performing multiple burns during the mission lifetime, which will create undesirable perturbations to instrument pointing to Alpha Centauri.

Another option is if ACESat were able to obtain a LEO that had minimal eclipses. A dusk/dawn Sun Synchronous Orbit (SSO, i = 97–99 deg) positions a satellite on the terminator where there will not be eclipses for nine months out of the year due to orbit precession. This orbit would allow for continuous stare time to Alpha Centauri during those nine months, however it requires a launch time of either 6 AM or PM and as a secondary, there is no control over the launch window.

Alternatively, if ACESat is stationed along the equator at a high enough altitude (1500 km) to just stare beneath the ecliptic for the entire mission, there would be uninterrupted access. However this orbit option is not commercially available.

DISCUSSION

The trajectories presented in this paper are analyzed for a secondary payload and are not exhaustive. Budget constraints of the SMEX proposal call limit the orbit possibilities for ACESat. While Earth orbits are preferred, they provide unstable thermal environments for ACESat. The best LEO option would be a dusk/dawn SSO, though as a secondary we are unable to specify the launch window. A heliocentric orbit, Earth Trailing or Leading, would provide a more beneficial mission for ACESat. The launch window variations for this orbit can be tolerated; by allowing RAAN to change ACESat can wait in a highly eccentric orbit to achieve a different RAAN value to escape from.

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